GENESIS HALO ORBIT STATION KEEPING DESIGN

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As the fifth mission of NASA's Discovery Program, Genesis is designed to collect solar wind samples for approximately two years in a halo orbit near the Sun-Earth L_1 Lagrange point for return to the Earth. The design of the maneuvers required for the station keeping in the halo orbits is described. An overview of the Genesis mission is provided with a brief description of the Genesis spacecraft and operational constraints, and a discussion of the contingency plans in the event of spacecraft or ground system anomalies.

1 - MISSION OVERVIEW

Genesis is the fifth mission selected by NASA's Discovery Program. Genesis is so named because the goal is to discover the origin of the formation of the Solar System. It is believed that the primordial material of the solar nebula remains today, in the interior of the Sun. To understand the origins of the Solar System, samples of this primordial dust must be obtained. But, how might this precious star dust be accessed inside the Sun? Fortunately, the Sun will actually deliver it to us – in the form of the solar wind. The neighborhood very near the Earth is, in fact, shielded from the solar wind by the Earth's magnetosphere. But beyond the Moon's orbit, outside the magnetosphere, approximately 1.5 million km from the Earth and in the direction towards the Sun, is the dynamically rich region near the L_1 Lagrange point. This is the ideal location for collection of the solar wind samples. The Genesis Project has therefore selected a large amplitude halo† orbit in the vicinity of L_1 such that the spacecraft can collect solar wind samples. The precious cargo is then returned to Earth at UTTR (Utah Test and Training Range) near Salt Lake City. Genesis will be the first NASA mission to return extraterrestrial samples back to the Earth since the lunar missions of the Apollo era. The Genesis mission design is more completely described by Lo et al. [Lo 98]

Besides its significant scientific value, Genesis is also the first mission designed with modern dynamical systems theory (Howell, Barden, and Lo [Howe 97], and Barden, Howell, Wilson, and Lo [Bard 97]). The baseline Genesis trajectory is unique in many respects. The most spectacular feature is the fact that after launch, only a single deterministic maneuver (ΔV) is required for the entire trajectory (for insertion into the halo orbit). At the minimum, this ΔV is only 6 m/s! Thereafter, the spacecraft completes four revolutions in the L_1 orbit before returning to the Earth — without any additional maneuvers. The Genesis trajectory appears in Fig. 1, plotted in the rotating frame that is standard for the three-body problem. The x-axis is parallel to the Sun-Earth line, directed from the Sun (at left) to the Earth (at the origin) . The xy-plane is the Ecliptic plane. The fully three-dimensional nature of the halo orbit is clear in the yz-projection. Examination of the xz-projection suggests the origin of the name "halo orbit": Farqhuar [Farq 80] so named it because it looks like a halo around the Sun, as viewed from the Earth.

† Techncially, the Genesis orbit is a quasiperiodic trajectory known as a Lissajous orbit, which is similar in nature to the type of orbit flown by ISEE3, the first libration point mission [Farq 80]. In fact, precisely periodic halo orbits only exist in the circular restricted problem. However, the term "halo orbit" generally includes such variations when the in-plane and out-of-plane amplitudes correlate closely to those associated with a periodc orbit.

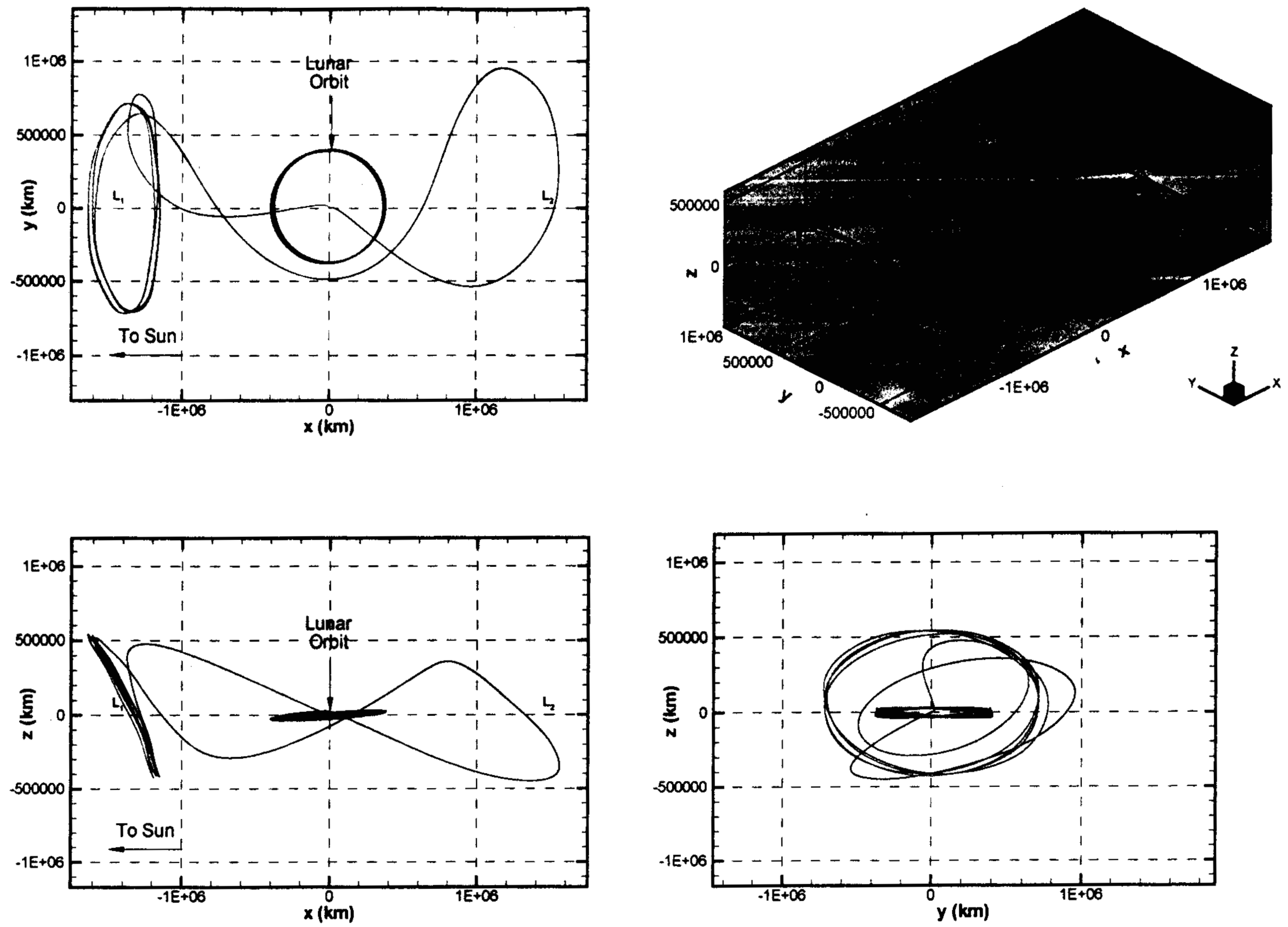
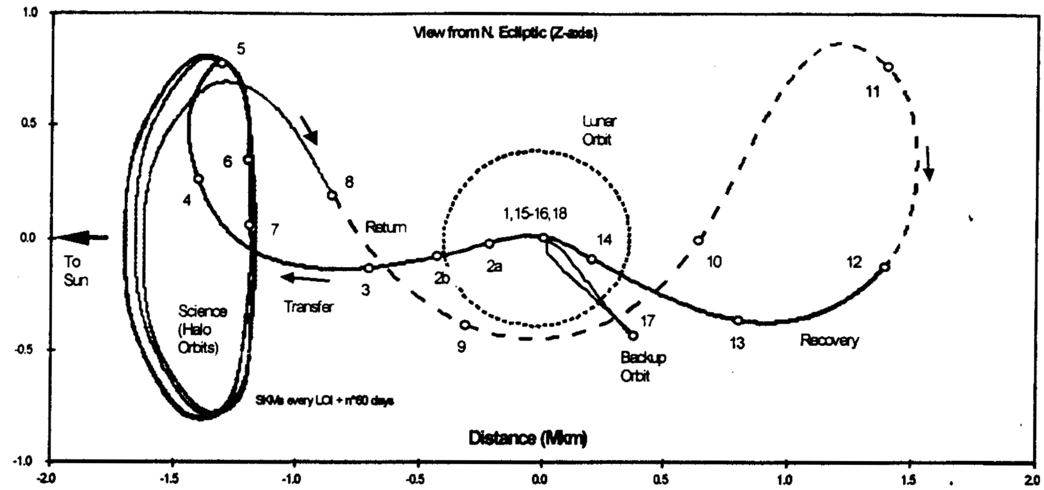


Fig. 1: Genesis Mission Trajectory Projections

Recall that the baseline trajectory includes no deterministic maneuver beyond halo orbit insertion. A second distinctive feature of the trajectory, is the return loop around L_2, on the far side of the Earth and away from the Sun. This design feature enables a dayside return to UTTR, otherwise unreachable via a direct return from L_1 orbits. The L_2 loop itself appears very similar to a partial halo orbit, and such a characteristic is no accident. The design exploits the dynamical channels, called heteroclinic connections, between L_1 and L_2 halo orbits (Howell, Mains, and Barden [Howe 94]). Inspired by Genesis, Koon, Lo, Marsden, and Ross [Koon 00] demonstrated that the region around the Sun-Earth system is riddled with these dynamical channels that serve as a source of chaotic motions in the Solar System. In fact, the entire Solar System is interconnected by a complex network of these dynamical channels (Lo and Ross [Lo 97]).

Of course, the launch and transfer to the L_1 orbit is also critical to the design. Launch opportunities are available in both January and February of 2001 on a Boeing Delta II 7326 launch vehicle with a Star 37 third stage providing a C_3 of -0.6 km^2/s^2. By the traditional calendar, Genesis will be NASA's first mission to be launched in the New Millennium. As many as four TCM's (Trajectory Correction Maneuvers) are planned to correct the launch error, as indicated in Fig. 2. Depending on the launch date, a 90 to 100 day transfer delivers the spacecraft to the halo orbit, where the LOI (Lissajous Orbit Insertion) maneuver inserts the spacecraft into the halo/Lissajous orbit. Shortly after LOI, the science mission begins for a minimum of 22 months of solar wind sample collection. About 13 SKM's (station keeping maneuvers) are anticipated to maintain the L_1 libration point orbit. After four revolutions, the spacecraft returns to Earth, sweeping around L_2.

A 19-day backup orbit offers a second entry possibility, should conditions be unfavorable for recovery on the first attempt. Parachute recovery by helicopters — in daylight — at UTTR concludes the space segment of the mission around August 2003.



KEY EVENTS (by Date/Nominal Mission Day):

1	Launch	07jan01*/000
2a-2b	TCM-1	08-10jan01*/001-003
3	TCM-2	14jan01*/007
4	TCM-3	11feb01*/035
5	Start Science	20mar01/072
6	TCM-4	20apr01/103
7	LOI (TCM-5)	30apr01/113
8	End Science	23mar03/805
9	TCM-6	09apr03/822
10	TCM-7	24apr03/837
11	TCM-8	13jun03/887
12	TCM-9	20jul03/924
13	TCM-10	09aug03/944
14	TCM-11	18aug03/953
15	Entry	19aug03/954
	--- OR ---	
16	Divert	19aug03/954
17	Apogee	29aug03/964
18	Backup Entry	07sep03/973

* Events that remain relative to launch opportunities (07jan01-20jan01 and 09feb01-20feb01); all other events delayed about one month for February launch.

Fig. 2: Maneuvers for Genesis Trajectory Correction (January 2001 Launch)

2 – SPACECRAFT DESIGN AND CONSTRAINTS

To achieve a level of cost-effectiveness consistent with a Discovery-class mission, the Genesis spacecraft design is adapted to the maximum extent possible from designs used on earlier missions, such as Stardust, another sample return mission. The spacecraft consists of a bus, including two solar arrays deployed after separation from the launch vehicle, and a Sample Return Capsule (SRC) with the science payload. The spacecraft is shown from two perspectives in Fig. 3.

Power is provided mainly by the solar arrays, with a 16 amp-hour battery in reserve. To avoid battery depletion, a time limit of about 85 minutes is imposed during which the spacecraft can be more than 30 degrees off Sun. Spin stabilization serves as a simple means of attitude control, in lieu of three-axis stabilization, with only a star tracker and two types of Sun sensors, but no reaction wheels, gyros, or accelerometers. To minimize contamination of the solar wind samples, thrusters are located only on the aft side of the spacecraft. The thruster configuration includes four 22 Nt (Newton) thrusters directed along the +X spacecraft axis and eight 1 Nt thrusters canted at 45 degrees away from +X towards +Z or −Z, which form part of a hydrazine fueled blowdown system. Thrusters are unbalanced, thereby introducing spurious ΔV's during each spacecraft pointing maneuver or spin rate change. Thruster activity, asymmetric mass properties, and misalignments also induce wobble and nutation and can result in maneuver execution errors as large as 6% in many cases.

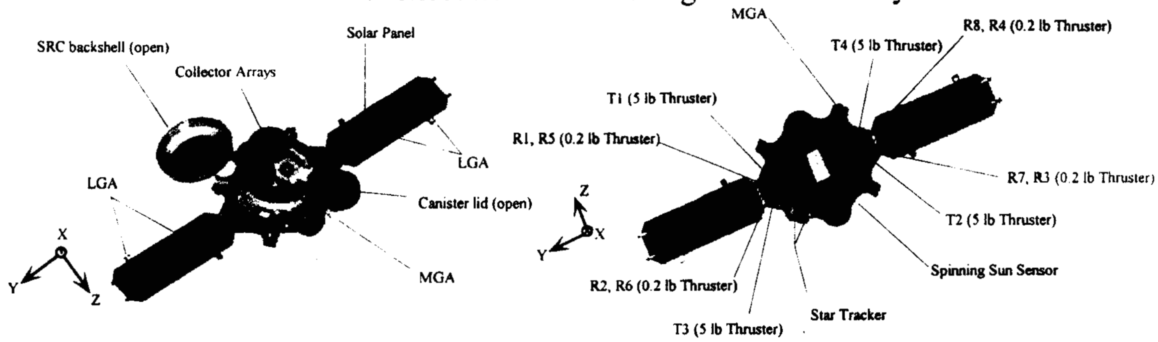


Fig. 3: Genesis Spacecraft Forward View (Normally Pointing Toward Sun) and Rear View

During solar wind collection, the SRC backshell is open and various science collection instruments are deployed. At these times, in order to maintain the attitude to collect the solar wind samples (with the spin-axis 4.5±0.5 degrees ahead of the Sun near the Ecliptic Plane), a maneuver may be required every day to compensate for the approximate one degree precession angle drift between the spin-axis and the Sun-spacecraft line . This does not affect the science collection, however, it may be necessary to interrupt the collection process to implement SKM's. Thermal requirements for instrument operations prevent the spacecraft from pointing more than 60 degrees away from the Sun during instrument operations to avoid irreparable damage to the primary science instrument, known as the Concentrator. If situations occur during which attitude adjustments of this magnitude are unavoidable, two options exist: to close the concentrator lid and SRC backshell; or, to shade the Concentrator with other deployable collection arrays. Both alternatives create an undesirable interruption in solar wind collection resulting in degradation of the quality of samples to be returned to Earth. Additionally, in the event that the required velocity correction exceeds 2.5 m/s, the 22 Nt thrusters are usually necessary. Procedurally, this requires an increase in the spin rate from the normal 1.6 RPM to 10 RPM to minimize nutation during the burn. As such, the SRC backshell must be closed with all science instruments in a stowed configuration whenever the larger thrusters are employed; this strategy also ensures the proper mass balance and spin characteristics.

Together, these constraints present a formidable challenge to mission design and navigation. To minimize interruptions to solar wind collection, it is desirable to use only the smaller thrusters and to keep maneuvers directed generally sunward for the various SKM's. Therefore, as a primary means of avoiding situations where the spacecraft must be oriented away from the Sun, all stationkeeping maneuvers must be biased towards the Sun. This approach will be discussed further in the following sections. Additional information regarding spacecraft design and related operational constraints is available [Will 99 and Will 00].

3 - STATION KEEPING AND THE DEVELOPMENT OF A BIASING STRATEGY

While other libration point missions have been flown successfully [Farq 85 and Shar 95], the Genesis spacecraft presents new challenges in many aspects of the overall mission. Of particular interest here are the constraints that affect the station keeping. The primary issue centers on the timing of the maneuvers in the Lissajous portion of the trajectory. In other libration point missions [Farq 85 and Shar 95], station keeping maneuvers were implemented as required resulting in an average frequency of two maneuvers per revolution (the revolution is defined as viewed in the projection onto the xy-plane), i.e., approximately one station keeping maneuver every three months. With this empirical data, as well as other studies and simulations of similar libration point trajectories, it is reasonable to first consider performing maneuvers at this frequency. However, because of the arrangements for tracking availability, the timing of the maneuvers is typically established at least one year prior to maneuver execution. In specifying the times of the maneuvers this far in advance, a degree of flexibility is removed from the station keeping algorithm, one which subsequently jeopardizes the ability to reliably maintain the trajectory in the face of uncertainties in the system (in the form of orbit determination errors, maneuver execution errors, etc.).

Early in the mission planning process, a second type of issue also surfaced that significantly constrains the maneuvers. Due to hardware constraints and operational complexities, the feasibility of designing all maneuvers such that each would possess a magnitude of at least 1 m/s was investigated. Additionally, all maneuvers were to be implemented such that the $\Delta\bar{v}$ vector is in the direction of the Sun along the Sun-spacecraft line. Ideally, the combined effect of both of these issues would keep the spin axis of the spacecraft within 30 degrees of the Sun-spacecraft line (arising from operational constraints). Each issue was investigated [Howe 98 and Bard 98] and significant results are summarized here.

3.1 - Station Keeping Strategy

The first step in the analysis of the station keeping issues for the Genesis mission requires an evaluation of a number of different control options to determine which type of controller is best suited to the constraints of the mission. While many options exist and have been tested, a simplified version of a target point method developed by Howell and Pernicka [Howe 93] is used to generate the results presented here. The goal of this controller is simply to apply an impulsive maneuver, or ΔV, in order that the error, the deviation in position between the actual trajectory and the nominal trajectory at some target time downstream, is reduced to zero. This is a tighter control philosophy than is used in other missions. Due to the criticality of the necessary timing in the trajectory for the return toward the Earth and subsequent reentry, it is necessary to more closely follow the nominal trajectory. Let $\Phi(t_1,t_0)$ be the STM (state transition matrix) associated with the nominal trajectory from the time of the maneuver (t_0) to the time corresponding to the target location along the nominal trajectory (t_1). This matrix is partitioned into four 3x3 submatrices as follows:

$$\Phi(t_1,t_0)=\begin{bmatrix} A & B \\ C & D \end{bmatrix} \tag{3.1}$$

The control can then be evaluated as:

$$\Delta\bar{v} = -B^{-1}A\bar{p} - \bar{e} \tag{3.2}$$

where $\bar{p}$ and $\bar{e}$ are the vector deviations from the nominal trajectory in position and velocity, respectively, at the time of the maneuver. This targetor results in a one-step computation of the required maneuver. Note that this *computation* is based on a linear propagation of the current state errors to the target time via the STM associated with the nominal solution. Since the actual propagation is performed using numerical integration of the full nonlinear differential equations, a very small residual error at the target will result. However, this method has been compared against a more exact method, and both provide essentially identical results in terms of control energy, thus rendering the residual error negligible within the framework of this investigation. *Note also that this method of computation is consistent with other targeting methods used in other types of maneuver and trajectory analyses software.*

To develop a strategy for station keeping maneuvers in the Lissajous portion of the trajectory, several scenarios are investigated in Howell and Barden [Howe 98] that include different levels of uncertainty in the execution errors, along with various strategies for frequency and location of the planned station keeping maneuvers. The results that are presented here represent a subset of the original study. Specifically, only the testing conditions that include the most severe execution errors (considered a worst case scenario at the time of the study) are summarized here. Those test conditions include a simulation duration of 680 days beyond the Lissajous orbit insertion (a time frame that extends over the entire interval in the Lissajous orbit); a maximum allowable maneuver of 2.5 m/s; orbit determination errors with 1 σ values of 3.96, 2.82, and 10.65 km for position and 8.46, 6.44, and 11.1 mm/s, respectively, for velocity (all components in rotating libration point coordinates); injection errors with the same 1 σ values as the OD errors; proportional execution errors with 7% (1 σ) downtrack and 4% crosstrack (per axis); and fixed execution errors of 0.02 m/s (3 σ) fixed error (per axis). In what is termed "configuration B," a two maneuvers per revolution strategy is employed where the maneuvers are specified to occur near the rotating xz -plane crossing. The target time for any maneuver coincides with the next planned maneuver time (except for the last maneuver that targets the end of the Lissajous portion of the trajectory).

The first critical measure of the effectiveness of this configuration (i.e., two maneuvers per revolution) is the number of cases (out of the 1000 simulated) that diverge. For numerical application, "divergence" is defined as the condition that a spacecraft state vector is beyond the vicinity of the nominal trajectory and cannot be targeted back to the nominal trajectory within the framework of the established testing conditions. Specifically, the spacecraft has drifted far enough from the nominal trajectory that a maneuver larger than the maximum allowable maneuver is required, or some other multiple maneuver targeting scheme is necessary to return the spacecraft state to the nominal solution. Ideally, complete assurance is sought such that the stationkeeping scheme can manage all possible errors (within the framework of the modeled errors) and avoid even a single divergence. In fact, this is not unreasonable to expect. However, for configuration B, a total of 825 cases out of 1000 diverge. The cause of this surprisingly high number of divergences can be linked to the two maneuver per revolution strategy with the maneuvers occurring at pre-specified times. Clearly, allowing larger maneuvers could help (although even with a 20 m/s maximum allowable maneuver, there are still more than 500 diverging samples), but this is not a practical solution. (It is acknowledged that constraining the timing of the maneuvers is a significant factor.)

The only flexibility available in the testing conditions is in the frequency of maneuvers. Thus, two new configurations, termed configurations C and D, are considered where three and four maneuvers per revolution (respectively) are utilized. In each configuration, the maneuvers are distributed evenly in time. Consistent with the previous case, the maneuvers are executed at specified times. In both configurations, there are no divergences among the 1000 cases simulated for each configuration. Clearly, this is a tremendous improvement. In this situation, it now becomes more meaningful to consider associated costs for the different configurations. For configuration C, the average cost for one case is 1.75 m/s with costs ranging from 0.59 to 9.02 m/s. Configuration D, not surprisingly, has a lower average cost of 1.13 m/s and a much narrower range from 0.53 to 2.09 m/s. Given the size of the execution errors, both of these configurations must be considered to be reliable options for the station keeping scheme to eventually be employed.

3.2 – Biasing Strategy

The other significant issues in establishing a station keeping scheme for the Genesis mission are the direction and magnitude of the maneuvers. Recall that all maneuvers are specified with a magnitude of at least 1 m/s and that all maneuvers are to be directed toward the Sun. Before attempting to resolve these matters, it is necessary to understand the characteristics of the maneuvers corresponding to the configurations defined previously. When simulated with a 1 σ value of 2.5% proportional execution error per axis instead of the values given previously (one of the possible sets of error levels at the time of the original analysis), all maneuvers for configuration C are less than 0.5 m/s while the maneuvers for configuration D are all less than 0.3 m/s. The directions of the maneuvers are evaluated in terms of the angle between the computed $\Delta\bar{v}$ vector and the Sun-spacecraft line. It happens that the distribution over all maneuvers for this angle is relatively even for both configurations. Specifically, this implies that the required maneuver is as likely to be orthogonal to the Sun-spacecraft line as it is to be directed toward or away from the Sun.

The problem then can be simply formulated as follows: how can a strategy be developed so that *all* station keeping maneuvers are directed toward the Sun, *all* maneuvers are at least 1 m/s in magnitude, and is robust enough that maneuver times can be specified a minimum of one year in advance? One answer is to bias the nominal trajectory with some specified number and magnitude of deterministic maneuvers in the direction of the Sun; the station keeping maneuvers are then overlaid on those deterministic maneuvers. The magnitude of the deterministic maneuvers is selected so that any computed station keeping maneuver that is added to the deterministic maneuver will not drive the magnitude of the resultant $\Delta\bar{v}$ to a value less than 1 m/s. Of course, this magnitude is primarily a function of the number of maneuvers per revolution. For example, if there are four maneuvers per revolution, the size of a station keeping maneuver will be less than 0.3 m/s. In this case, the strategy would call for four deterministic maneuvers per revolution in the negative rotating libration point (RLP) x-axis (a close enough approximation to the negative direction along the Sun-spacecraft line) per revolution, equally spaced in time, each with a magnitude of 1.3 m/s. Similarly, in the case of configuration C, the trajectory is redesigned with maneuver biases of 1.5 m/s that coincide with the station keeping opportunities. This, along with several other scenarios, has been simulated [Bard 98].

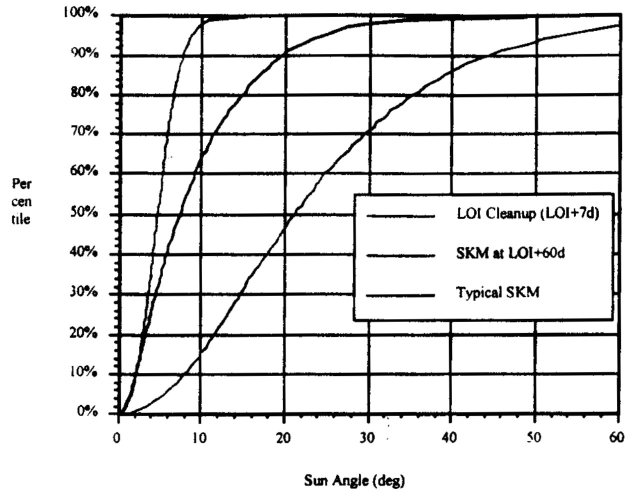
The results from this strategy are very convincing. Across all simulations, various combinations, that include the frequency of the maneuvers and the magnitudes of the resulting deterministic maneuvers, have been found that keep all of the maneuvers in a direction that is no more than 9 or 10 degrees off of the Sun-spacecraft line. Additionally, the variation in the total cost is small. For configuration C, the average cost over the duration of the simulation, including the 1.5 m/s biases, is 18.04 m/s with costs over a range from 16.56 to 19.28 m/s. Similar to the trends seen previously, the variation in costs narrows further for configuration D ranging from 18.64 to 19.84 m/s with a 19.25 m/s average. This predictability in the cost in the presence of the types of parameter uncertainties is appealing.

4 - VERIFICATION OF SKM BIASING STRATEGY

As a check on the feasibility of the three-maneuver per orbit strategy recommended above, and to establish a biasing level which is operationally robust, software called LAMBIC (Linear Analysis of Maneuvers with Bounds and Inequality Constraints) is used to perform monte-carlo simulations for numerous scenarios. About 5000 samples are selected based on maneuver execution errors of 6% in the direction of the nominal maneuver and 4% in each orthogonal direction (each 3 σ). The impact of all previous maneuvers is also modeled, including launch injection errors. A deterministic bias level of 1.5 m/s is selected based on the studies summarized in the previous section; conveniently, 1.5 m/s falls half way between the typical two-way turn circle diameter (0.5 m/s) and the normal maximum maneuver size on the 0.2 lbf thrusters (2.5 m/s) [Will 99 and Will 00].

The comparison in Fig. 4 indicates performance in terms of total Δv direction and magnitude, as obtained from LAMBIC, for the following cases, all with a nominal bias level of 1.5 m/s:

- Largest SKM expected (post-LOI cleanup maneuver)
- Smallest SKM expected (60 days after the LOI maneuver)
- Typical SKM



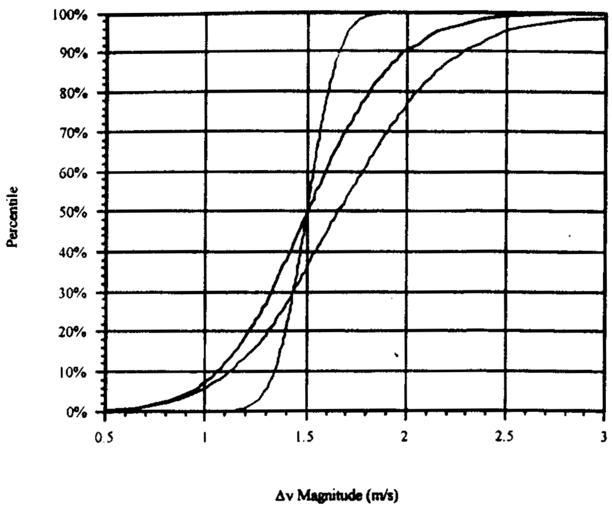


Fig. 4: Comparison of Performance for Halo SKM's Biased at 1.5 m/s

These studies suggest that a combination of the strategy of three station keeping maneuvers per halo orbit with sunward biasing of 1.5 m/s per maneuver provides a feasible approach to meet mission objectives, while satisfying spacecraft and instrument operational constraints. The strategy is robust, both in terms of staying within the 0.5-2.5 m/s range, as well as maintaining pointing to within 60 degrees of the Sun in the vast majority of cases. Note that the actual direction of the burn is slightly different than the ΔV direction indicated in Fig. 4, because of the effect of maneuver decomposition. Over the course of the four halo orbits, with execution errors considered, the effect of biasing adds about another 30 m/s to the mission ΔV budget. This is easily covered by the available margin of 67 m/s.

However, what if an anomaly occurs which prevents an SKM from being executed according to schedule or gives rise to a large spurious ΔV? Such contingency planning as the final element of the overall station keeping strategy is addressed in the next section.

5 - CONTINGENCY PLANNING

Ultimately, the goal of the Genesis Mission is to return its solar wind samples back to UTTR for mid-air retrieval by helicopter. This goal is the predominant constraint in the design of the nominal trajectories for both the January and February 2001 launch opportunities. It is also the primary design driver for any contingency plans that may be utilized. In the end, the SRC *must* be delivered to an acceptable Entry Interface Point (defined at a specific altitude, latitude, longitude, and flight path angle) in order to enter the Earth's atmosphere with the proper conditions that ensure the SRC reaches the recovery zone.

A series of contingency plans, with varying orders of complexity, are currently being examined to ensure recovery of the SRC at the end of the mission. A summary of these plans is presented next, in order of complexity from simplest (and most common) to most difficult. The first two involve targeting back to the reference trajectory and are:

- Design the current SKM or TCM by targeting to the following SKM/TCM opportunity. This is the baseline targeting scheme discussed in this paper and assumes that the final target is determined from the reference trajectory. The reference trajectory itself is unchanged for this type of solution.

- Base the current SKM/TCM on a target further downstream, but still on the reference trajectory, and allow for additional maneuvers at various intermediate points. For instance, in the design for SKM-3A, it may be advantageous to target SKM-4A (one rev later), while allowing intermediate maneuvers at SKM-3B or SKM-3C, as well as, SKM-4A. This plan allows the correction back to the reference to be spread out over a series of corrective maneuvers, while leaving the reference trajectory unchanged after the final target (in this case after SKM-4A).

The next set of contingency plans develop a new reference trajectory on which to base the target for the current SKM/TCM. This can cover a number of different scenarios that can be divided into three broad categories:

- Redesign the reference trajectory from current location to some point on the original reference solution. A prime example of this would be a redesign early in the trajectory to target through the entire halo to the "Return Maneuver" at TCM-6 (see Fig. 2). This would introduce a deterministic maneuver at TCM-6, but would preserve the critical portion of the trajectory after TCM-6 leading back to entry. All SKM/TCM's prior to this location would target the new reference trajectory, while those after would target the original.
- Redesign from current location to a new Entry Interface Point. This involves a complete change to the reference solution and would presumably only be investigated very late in the mission, or if the correction maneuver in previous redesign becomes prohibitively large or impossible to implement. This type of contingency plan would alter the quantitative nature of the entire reference solution, but would not change the qualitative characteristics exhibited in Fig. 1.
- The final contingency type would be to redesign from the current or even a future location to a vastly different reference trajectory that does not preserve the characteristics of the nominal trajectory. This contingency has not been explored in depth, but may involve, for instance, a series of lunar encounters and/or phasing loops to recover the SRC. This plan is viewed as a last resort to save an otherwise lost mission, and would be utilized only if all other options have been exhausted.

The design of contingency plans for the Genesis mission is at the same time fragile and yet robust. It is fragile in the sense that mission success is defined only by a successful delivery of the SRC back to UTTR using a very complex trajectory design space. However, this complex design space has proven itself to be very robust to additional constraints and design changes and there is confidence that, should the need arise, a suitable contingency plan can be developed to ensure mission success.

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